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TITLE- Implications of Lower Lunar Orbital Altitudes for LM Separation

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AUTHOR(S)- D. R. Valley

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Extended Lunar Missions
Lower Lunar Orbit
Altitude for LM Separation
Lunar Payload Capability

ABSTRACT

The concept of delivering the LM to a lower lunar orbit before separation from the CSM is investigated from several standpoints:

1. Increased payload capability
2. Service Module propellant requirements for different delivery techniques
3. Launch vehicle requirements
4. Communications
5. General operational problems

Landed payloads of the LM can be increased by as much as 840 lbs for a 50,000 ft altitude. Insertion directly into this low orbital altitude, however, would be risky, making an intermediate Hohmann transfer maneuver desirable.

Further study is required to determine the minimum altitude to which the direct retro can be safely made and to evaluate the problems associated with LM ascent and rendezvous at the lower altitudes.

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BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: Implications of Lower Lunar Orbital Altitudes for LM Separation - Case 232

DATE: August 1, 1967

FROM: D. R. Valley

TM-67-1012-8

TECHNICAL MEMORANDUM

I. INTRODUCTION

Recent AAP lunar mission planning efforts suggest the concept of delivering the LM to a lower lunar orbit before separation from the CSM. The prime reason for the lower altitude is to reduce the ΔV requirements for the LM descent and ascent maneuvers. For a given LM descent propellant tank capacity, the reduced ΔV requirements will allow greater LM separation weights and thus increase the landed and returned payload capabilities.

Greater LM capability, however, comes at the price of increased SM propulsion (SPS) requirements. The heavier LM places greater demands on the SPS, as do the increased ΔV requirements necessary to deliver the LM to a lower lunar orbit. In general, the concept of lower altitudes redistributes the mission ΔV requirements. Taking advantage of the lowered LM ΔV budget will always be accompanied by increased demands on the SPS, and it becomes a question of balancing these shifts in mission requirements to arrive at the spacecraft configuration that will produce the greatest landed payload increase within existing equipment capabilities; that is, within existing spacecraft tank capacities and launch vehicle injection capability. In addition, guidance, navigation, and operational problems in the lower orbit must be considered.

II. MAXIMUM ALLOWABLE LM SEPARATION WEIGHT

A lower orbital altitude at the time of LM separation will reduce the LM ΔV budget for both descent and ascent. Figure 1 is a plot of the reduction in LM ΔV requirements for altitudes lower than 80 nm.

Using the existing usable propellant tank capacity of the LM descent stage (17,441 lbs), a maximum LM separation weight can be simply calculated by applying these ΔV reductions to the nominal Apollo descent ΔV requirement (7,332 fps) as follows:

[redacted]

LM separation weight = (descent propellant weight) $\left(\frac{MR}{MR-1}\right)$

where MR is the mass ratio associated with

the descent ΔV required.

$$MR = \exp \frac{\Delta V_{Descent}}{g_o I_{sp}}$$

Figure 2 is a plot of the maximum allowable LM separation weight for altitudes down to 50,000 ft. The maximum allowable LM separation weight for 50,000 ft is 33,274 lbs, an increase of 685 lbs over the baseline Apollo LM (32,589 lbs). This difference in separation weight includes 45 lbs of additional descent propellant to completely fill the tanks and a 640 lb payload increase. This additional payload can be obtained while retaining a fully loaded ascent stage to permit a nominal Apollo ascent and rendezvous at 80 nm.

If LM ascent and rendezvous at the lower orbital altitudes is considered, the ascent ΔV budget is also reduced in accordance with Figure 1; and ascent propellant can be off-loaded to further increase the landed payload capability. Figure 3 is a plot of landed payload increase vs. orbital altitude at LM separation altitudes down to 50,000 ft. The lower line applies if the nominal Apollo type ascent and rendezvous to 80 nm is used, while the upper line indicates the increased payload obtainable by returning to the lower altitudes.

Figure 4 indicates the degree of flexibility possible with the existing Apollo LM tank capacities in terms of trading between increases in landed payload and payloads returned to lunar orbit. If the landed payload is reduced, ascent propellant can be added to increase the returned weights. Figure 4 illustrates the extent to which this trading can occur up to the point where the LM ascent tank capacity is reached.

III. SERVICE MODULE PROPELLANT REQUIREMENTS

Up to this point, only the improved LM performance has been discussed in connection with the lower altitudes. As mentioned earlier, taking advantage of the LM improvements possible with lower separation altitudes increases the requirement on the Service Module Propulsion System (SPS). The heavier LM and the maneuvers necessary to deliver it to lower orbital altitudes place a heavier demand on the Service Module.

IV. ΔV BUDGET

To evaluate the extent of these requirements, it was necessary to assume a nominal ΔV budget to work with. The following ΔV budget* was used to represent a nominal Apollo type mission with relaxation of the non-free return constraint and with 900 ft/sec allowed for CSM plane change capability to meet the anytime-abort constraint for the higher latitude landing sites:

ΔV for Transearth Injection = 3,190 ft/sec

ΔV for LM Rescue and Plane Change = 1,580 ft/sec (680 + 900)

ΔV for Lunar Orbit Insertion = 2,957 ft/sec

Many different AAP missions to specific sites have been studied and the ΔV requirements determined. The ΔV budget above was selected on the basis of being adequate to perform most of the missions studied. Since the budget leans to the worst case type missions, it should be considered to be on the conservative side.

V. MISSION MODES

The following means of delivering the LM to the lower orbital altitudes were considered:

I. Retro directly from translunar flight to the low circular orbit and depart for earth from the low altitude. This mode results in minimum SPS requirements.

II. Retro from translunar flight to an 80 nm orbit; Hohmann transfer to the low orbit, and depart for earth from the low orbit. This mode eases the translunar guidance and navigation requirements associated with direct retro to low altitudes.

III. Retro directly from translunar flight to the low orbit, Hohmann transfer up to 80 nm after LM separation, and depart for earth from the 80 nm orbit. This mode permits nominal Apollo rendezvous procedures.

IV. Retro from translunar flight to an 80 nm orbit, Hohmann transfer to the low orbit, Hohmann transfer back up to 80 nm after LM separation, and depart for earth from the 80 nm orbit. This mode eases the effect of translunar guidance and navigation errors as well as allowing the nominal Apollo rendezvous.

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*ΔV budget obtained from D. R. Anselmo, Bellcomm, verbal discussion.

The nominal ΔV budget assumed must now be adjusted to include the increased requirements imposed by these four modes. Hereafter, the mission modes will be referred to as I, II, III, and IV as numbered above.

Figure 5 indicates the ΔV penalties associated with direct retro into and departure from the lower orbits instead of the nominal 80 nm altitude. These increments must be added to the nominal ΔV budget for mission modes involving retro or departure from low orbits. The penalties associated with the Hohmann transfer maneuvers to lower orbits can be readily estimated from the difference in circular orbital velocities. For example, to perform a Hohmann transfer from 80 nm (V_c = 5,289 ft/sec) to 10 nm (V_c = 5,481 ft/sec) altitude requires a ΔV of 192 ft/sec for both burns.* Modifications to the nominal ΔV budget for each of the mission modes is shown in Appendix A which also indicates the derivation of the direct retro and departure penalties shown in Figure 5.

With the ΔV budgets determined for each of the individual mission modes, it is only a matter of standard weight performance calculations to determine the SM propellant requirement for each of the mission modes using LM separation altitudes between 80 nm and 50,000 ft. The method of calculating these requirements is shown in Appendix B. The results are plotted in Figure 6 which shows the SM propellant requirements vs. LM separation altitude for each of the mission modes.

Mission Mode I (direct retro and departure from the lower orbit) places the least requirements on the SPS system. It would be possible to deliver a 33,274 lb LM to a 50,000 ft orbit and complete the mission well within the present SM tank capacity. The landed payload for the mission could be increased by about 840 lbs (Figure 3).

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*The "rule of thumb" for estimating Hohmann transfer ΔV requirements can be accurately applied when the ratio of radii of the two circular orbits is near unity. For the case in point, i.e., transfer from an 80 nm to a 50,000 ft orbit, the ratio of radii is approximately 1.08 and error in ΔV is less than 0.1 ft/sec. (These ΔV requirements can be read from Figure 1, which is actually a plot of the differences in circular orbital velocities for the altitudes indicated.)

Mission Mode II (Hohmann transfer down from 80 nm) exceeds the SM tank capacity by some 850 lbs if the 50,000 ft altitude is considered. In order to complete a landing mission within the SM tank capacity, the LM separation altitude could only be reduced to about 34 nm. At this altitude, the maximum LM separation weight would be about 33,050 lbs (Figure 2) and the landed payload increase would be 540 lbs (Figure 3).

Mission Mode III (direct retro and Hohmann transfer back up to 80 nm) would enable delivery of the LM to a 20 nm separation altitude. The maximum LM separation weight would be 33,168 lbs and the landed payload could be increased by 535 lbs (Figure 3, lower curve). Although the LM separation weight is greater than for the previous case, the landed payload is slightly less. The LM ascent to 80 nm in this mission mode reduces the landed payload capability.

Mission Mode IV (retro to 80 nm, Hohmann transfer down and back up again after LM separation) would exceed the SM tank capacity by about 1600 lbs if a 50,000 ft LM separation altitude were desired. The existing SM tankage would limit the separation altitude to about 44.5 nm. The maximum LM weight for this altitude would be 32,955 lbs and the landed payload increase would only be about 325 lbs (Figure 3, lower curve).

The results plotted on Figure 6 provide a good feel for the effects of the different mission modes and the constraint imposed by the SM propellant capacity. Figure 7 presents a look at the launch vehicle injection capability required. A summation of SM propellant, LM separation weight (less crew), CSM weight, and the adapter for each of the mission modes is plotted on Figure 7. It can readily be seen that all mission modes exceed the 98,000* lb Saturn V capability for a 50,000 ft LM separation altitude. Mission Mode I, the least demanding from a weight standpoint, would require a launch vehicle capability of about 98,700 lbs. The baseline weights for the CSM (23,562 lbs) and the Adapter (3,850 lbs) were used throughout the mission calculations.

VI. SENSITIVITY TO ΔV AND INERT WEIGHT CHANGES

The mission weight analysis presented thus far has been based on the nominal ΔV budget which, admittedly, may be on the conservative side and possibly presents an unfair picture. The weight performance calculations, however, can be

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*Saturn V capability for non-free return missions should exceed the 98,000 lb capability guaranteed for Apollo.

turned around to provide information on the maximum allowable ΔV budget for the existing SM propellant capacity. The results of such an analysis are shown in Figures 8 and 9 and the method of obtaining the results is discussed in Appendix C.

All results shown in Figures 8 and 9 are based on using the baseline value for CSM inert weight (23,562 lbs). As the lunar mission durations are extended, the CSM weight will increase, The SM propellant requirement will go up from those shown on Figure 6, and the allowable ΔV budgets indicated on Figures 8 and 9 will be reduced. The point is that the results shown in this memorandum are highly dependent on the CSM inert weight and the timing associated with its expendable weight losses during a mission.

VII. OPERATIONAL CONSIDERATIONS

Thus far, little has been said about the general operational aspects of delivering a LM to a lower separation altitude. The following discussion will touch on some of the operational problems that can be foreseen for this type of mission.

Guidance and Navigation

The Apollo guidance system accuracy for lunar orbit insertion can introduce a 1σ altitude error of ±5,000 ft for the direct retro maneuvers of mission modes I and III. For a 50,000 ft orbital altitude, a possible 5,000 ft error along with the uncertainties in connection with the position, shape, and extent of surface features of the moon must be considered risky. When the possibility of an over burn of the SPS during the large retro maneuver is considered, the situation becomes even more uncertain. For example, the difference in orbital velocities at 50,000 ft altitude and at the lunar surface is only 24 ft/sec—not much margin for insertion errors.

One sigma altitude errors of the Apollo guidance system in connection with the smaller impulses of the Hohmann transfers of mission modes II and IV would only be ±500 ft. In addition, the position of the spacecraft can be accurately checked out in the higher parking orbit before the Hohmann transfer is initiated. A higher initial parking orbit also offers a better opportunity for navigational sightings to surface landmarks. These sightings would be an aid in obtaining the most advantageous LM separation orbit with respect to the desired landing site.

LM Ascent and Rendezvous

From the standpoint of propulsion savings, it is desirable to ascend and rendezvous at low orbital altitudes; however, there are some operational aspects that must be considered. The nominal Apollo procedures rely on different orbital altitudes for phasing of the CSM and LM ascent stage. There presently is a 5 1/2 minute lunar launch window allowable for the Apollo mission. If the CSM is parked in a 50,000 ft orbit, and rendezvous by an active LM at this altitude is assumed, the launch of the ascent stage would have to be very accurately timed. One possible solution is to launch the LM to a higher altitude than the CSM. This procedure could provide the proper phasing, but would increase the ascent propulsion requirements. With a LM-active rendezvous from above, visibility may be a problem with the lunar surface as a background. Other solutions would be to use the LM rescue propellant allocation to perform a CSM-active rendezvous, or to simply modify the CSM orbit for proper phasing.

LM ascent and rendezvous with a CSM in a low parking orbit will require considerable study effort to develop new procedures from those planned for Apollo.

RCS Propellant Requirements

The additional SPS burns associated with the Hohmann transfer maneuvers of mission modes II, III, and IV will increase the RCS propellant requirements for the mission.* The RCS tankage will have to be increased and the CSM inert weight will grow. The extent of the tankage penalties plus the added propellant load will increase the SM propellant requirements or else cut back on the mission capabilities of the system using Block II SM tankage.

Landing Radar Operation

Operation of the landing radar should not be affected. Altitude updates during the Apollo type approach are not initiated until 25,000 ft altitude is reached, and velocity updating is not initiated until the altitude is 15,000 ft. The landing approach phases need not be changed by the lower LM separation altitude.

Communications

The orbital altitude will affect the central angle (θ) beyond the limb that the spacecraft can travel before

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*RCS engines are used for ullage burns prior to SPS starts as well as for attitude control during thrusting.

communications are blocked by the moon (see sketch on Figure 10), and thus may change the longitude accessibility. Figure 10 plots the change in angle θ with respect to orbital altitude.

One other consideration is the reduction in time per orbit that the CSM is in view (horizon to horizon) of the landed LM. An 80 nm orbit provides about a 15 minute viewing time, while a 50,000 ft orbit would reduce this to less than five minutes. The angular rates of the CSM passing overhead would be increased by a factor of 10 to about 6.3°/sec. Angular rates of this magnitude are approaching the 7° slew rate limit of the rendezvous radar antenna. Therefore, any requirements for LM tracking of the orbiting CSM with the rendezvous radar may be restricted by the reduced viewing time and increased angular rates associated with lower orbital altitudes. CSM tracking of the LM via sextant sightings would be more difficult for the same reasons.

VIII. CONCLUSIONS

The landed and returned payloads of the LM can be increased by separation and rendezvous at orbital altitudes below the present 80 nm. The increase in landed payload can be as great as 840 lbs for a 50,000 ft altitude.

From the discussion thus far, it is apparent that mission mode I (direct retro and departure) is the least demanding from a propulsion standpoint. Operationally, however, it may prove too risky to retro directly into a 50,000 ft orbit, and it also introduces the necessity of new procedures for the LM ascent and rendezvous.

Mission mode II reduces the dangers of direct retro by paying an additional penalty for a Hohmann transfer down, but the problems associated with ascent and rendezvous at the low altitude remain.

Mission mode III has the hazard of the direct retro maneuver, but eliminates new ascent and rendezvous problems by having the CSM perform a Hohmann transfer back up to an 80 nm parking orbit.

Mission mode IV, the most conservative, relieves the direct retro hazards by means of a Hohmann transfer down, and allows the conventional ascent and rendezvous procedures by means of the CSM Hohmann transfer back up to the 80 nm parking orbit after LM separation.

The final selection of a mission mode should weigh the landed payload capability increases provided by lower orbital altitudes, the element of risk associated with direct insertion to these altitudes, and overall mission capability constraints imposed by the SM propellant requirements of the most conservative mode.

Two areas will require further analysis before a good selection can be made:

1. determination of the minimum altitude to which the direct retro can be made;

2. evaluation of the problems associated with ascent and rendezvous maneuvers to lower orbital altitudes.

If the two above problems were accurately and realistically evaluated, the best compromise mission mode could be selected. As an example, assume that ascent and rendezvous to 50,000 ft were shown to be feasible and that the lowest altitude for safely executing the direct retro maneuver was determined to be 40 nm. LM separation from and ascent to a 50,000 ft orbit would offer the best landed payload capability. Direct retro to 40 nm and a Hohmann transfer to 50,000 ft would prove the most efficient and safe means of delivering the LM to 50,000 ft. This, essentially, is mission mode II except that initial lunar orbit immediately after retro is 40 nm instead of 80 nm.

D. R. Valley

1012-DRV-hjt D. R. Valley

Attachments:
Figures 1 - 10
Appendixes A - C

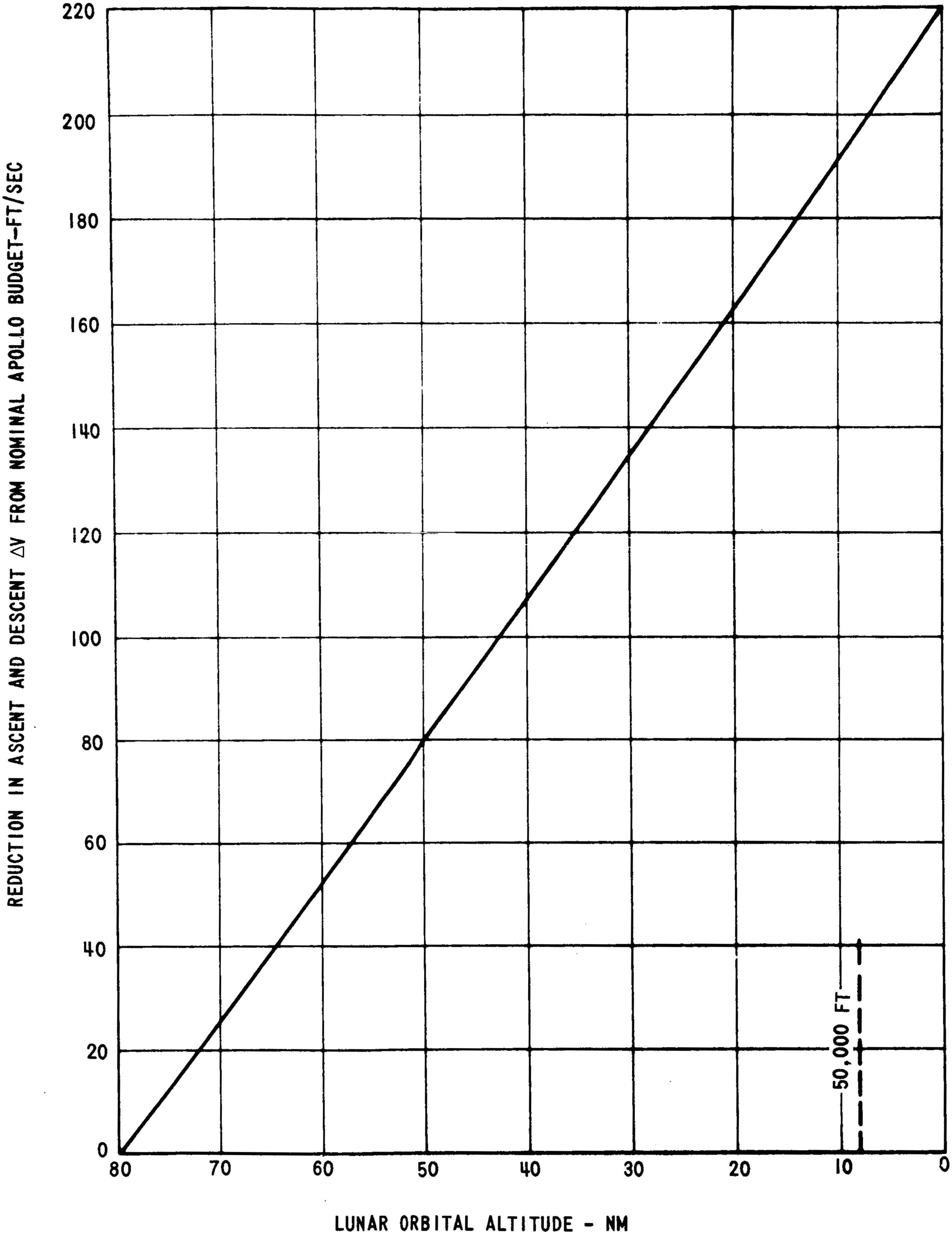

FIGURE 1 - REDUCTION IN LM ASCENT & DESCENT ΔV FOR LUNAR ORBITAL ALTITUDES BELOW 80 NM

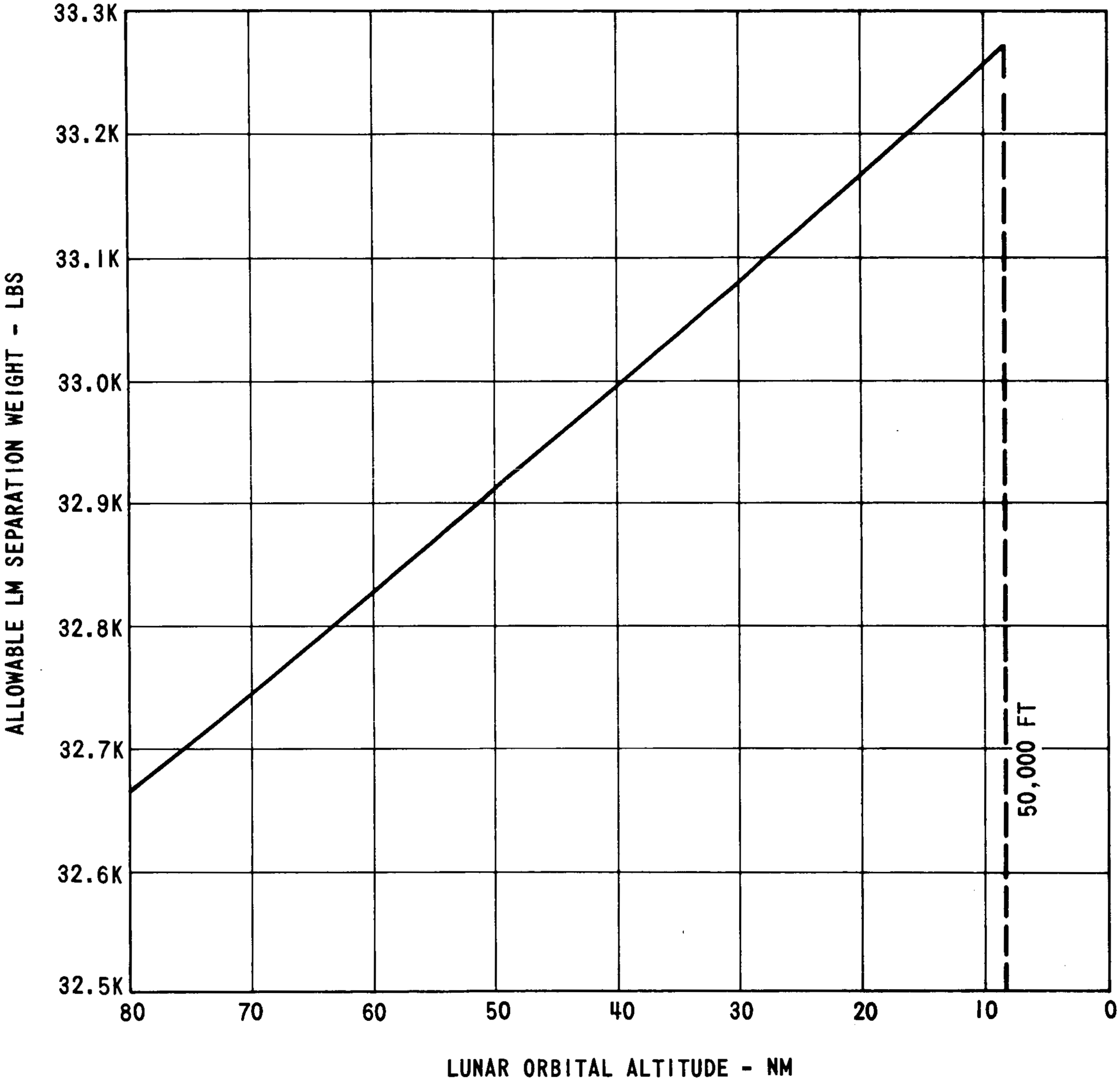

FIGURE 2 - ALLOWABLE LM SEPARATION WEIGHT vs. LUNAR ORBITAL ALTITUDE
(BASED ON FULL DESCENT PROPELLANT TANKS - 17,441 LBS)

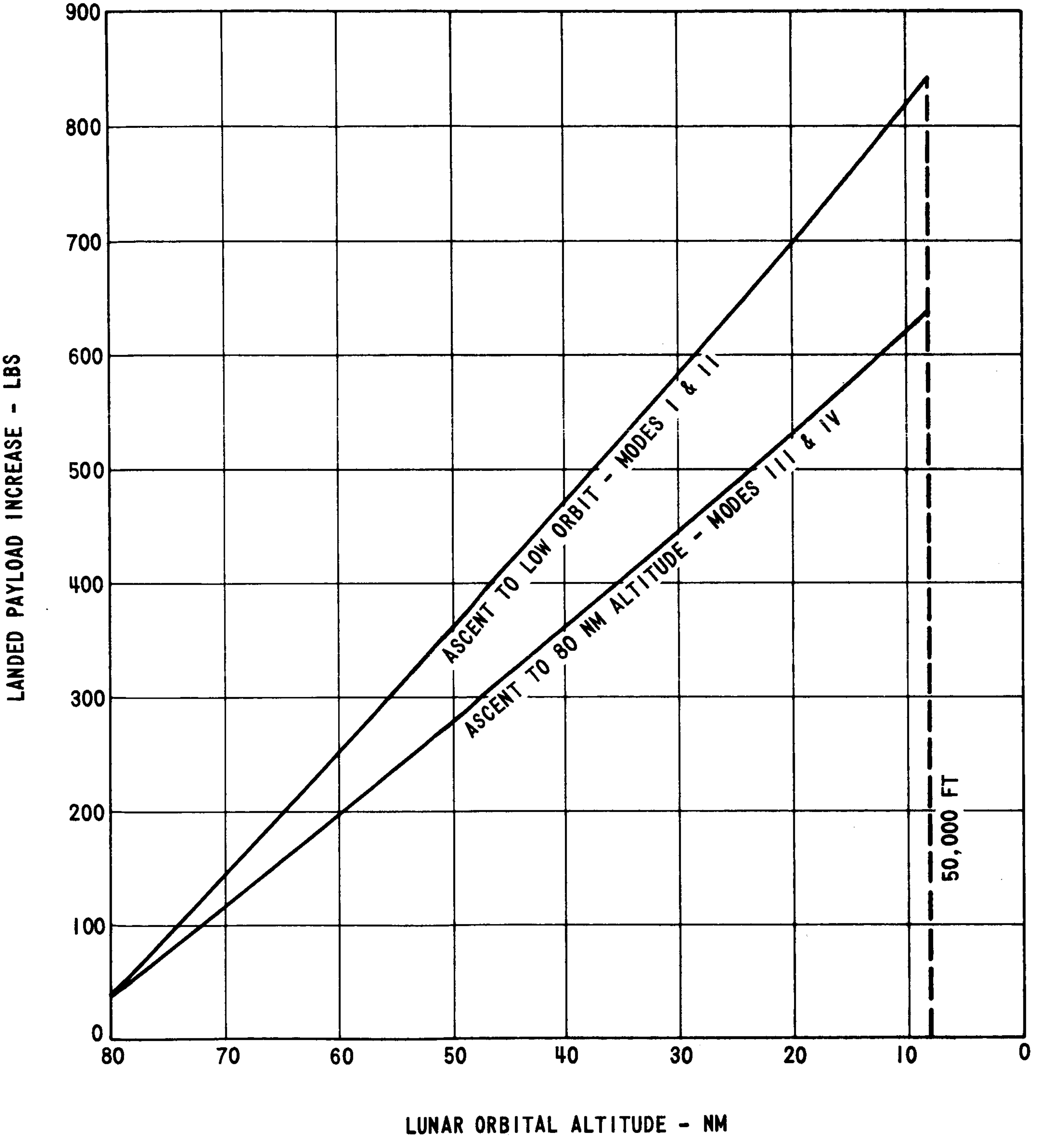

FIGURE 3 - LANDED PAYLOAD INCREASE vs. LUNAR ORBITAL ALTITUDE
(BASED ON FULL LM DESCENT TANKS - 17,441 LBS)

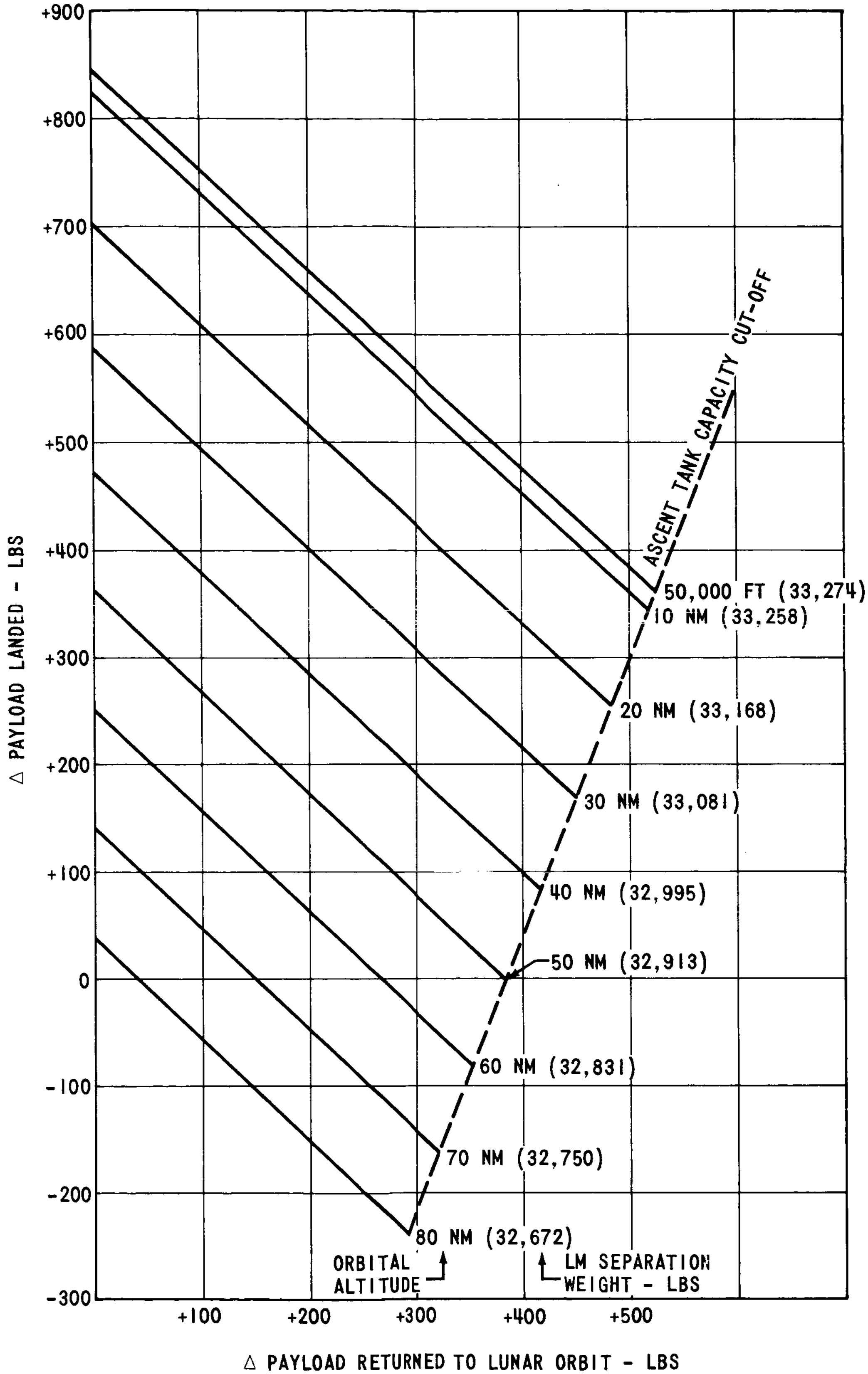

FIGURE 4 - POSSIBLE TRADES BETWEEN △ PAYLOADS LANDED AND △ PAYLOADS RETURNED TO LUNAR ORBIT WITH EXISTING LM TANK CAPACITIES

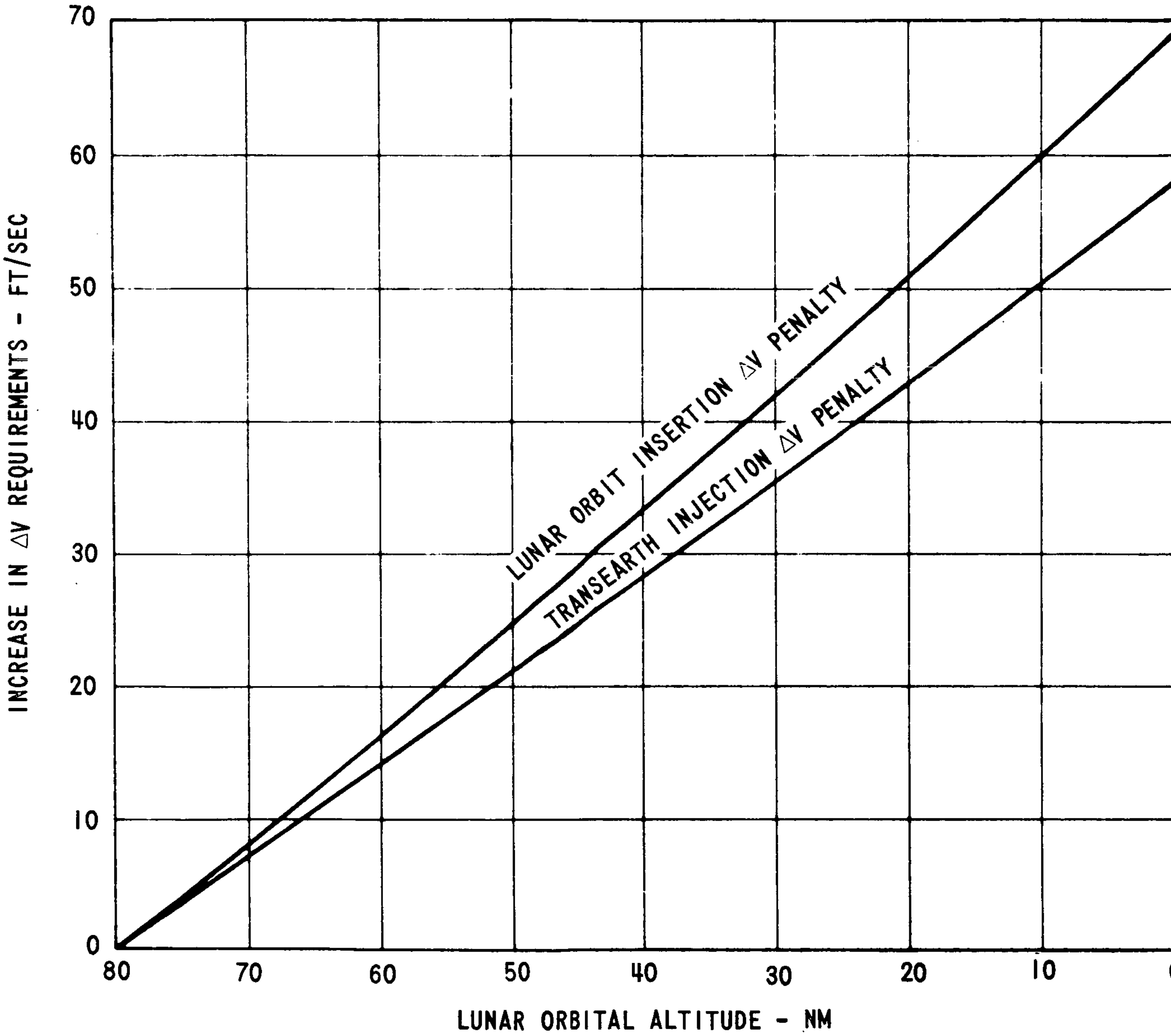

FIGURE 5 - INCREASE TO NOMINAL △V REQUIREMENTS FOR RETRO & DEPARTURE FROM LUNAR ORBITAL ALTITUDES LOWER THAN 80 NM

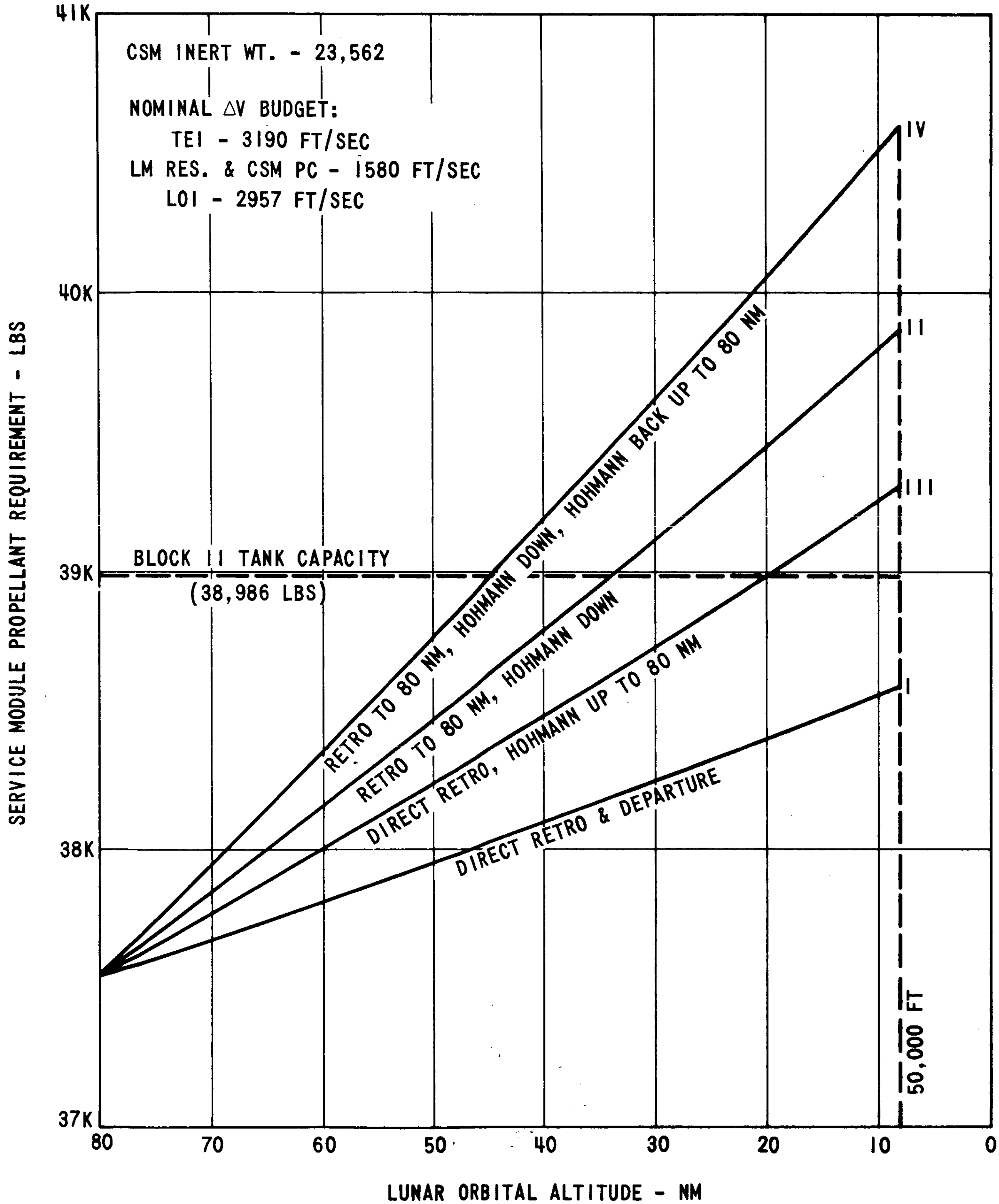

FIGURE 6 - SM PROPELLANT REQUIREMENTS FOR MISSION MODES I THRU IV

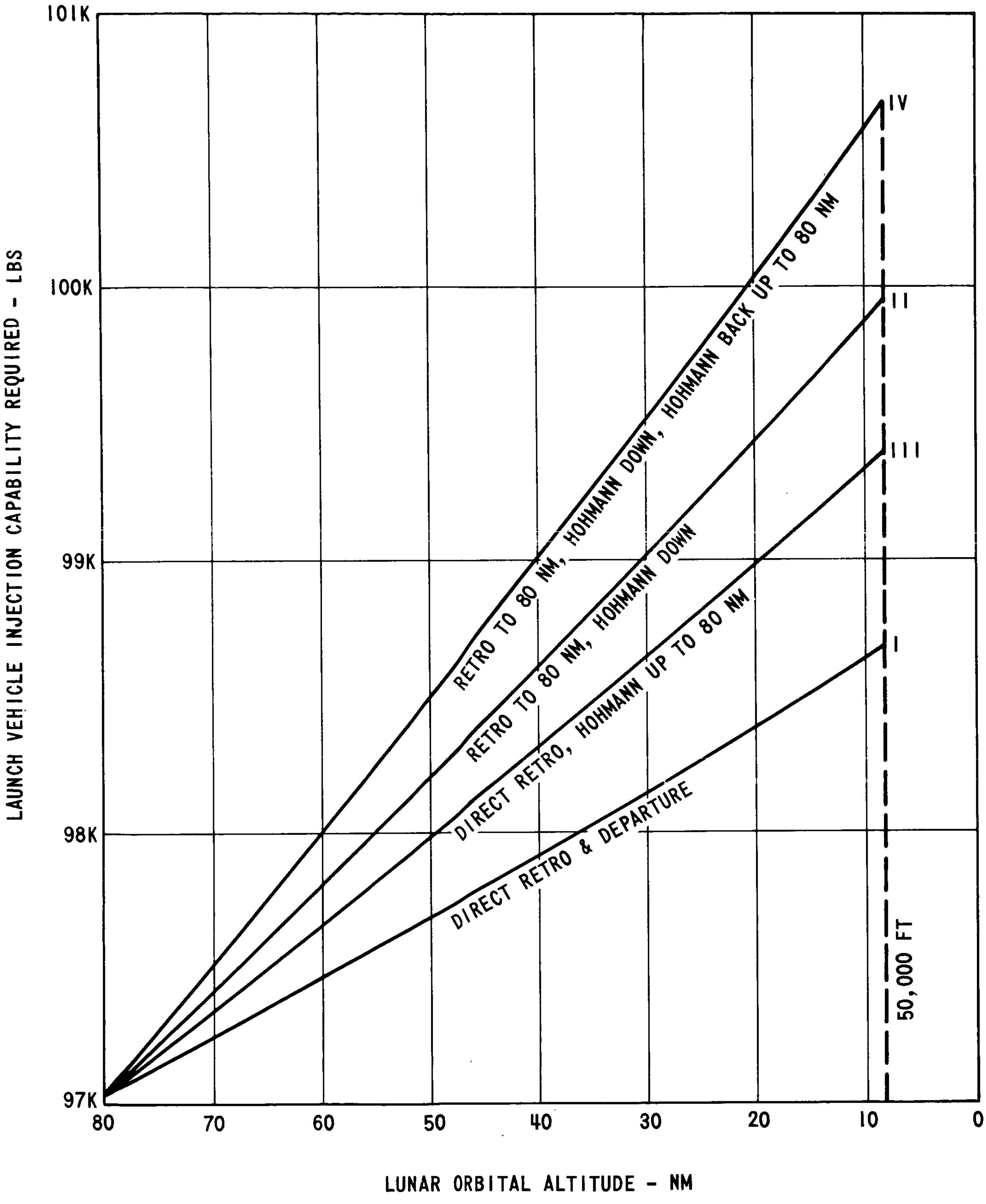

FIGURE 7 - LAUNCH VEHICLE INJECTION REQUIREMENTS FOR MISSION MODES I THRU IV

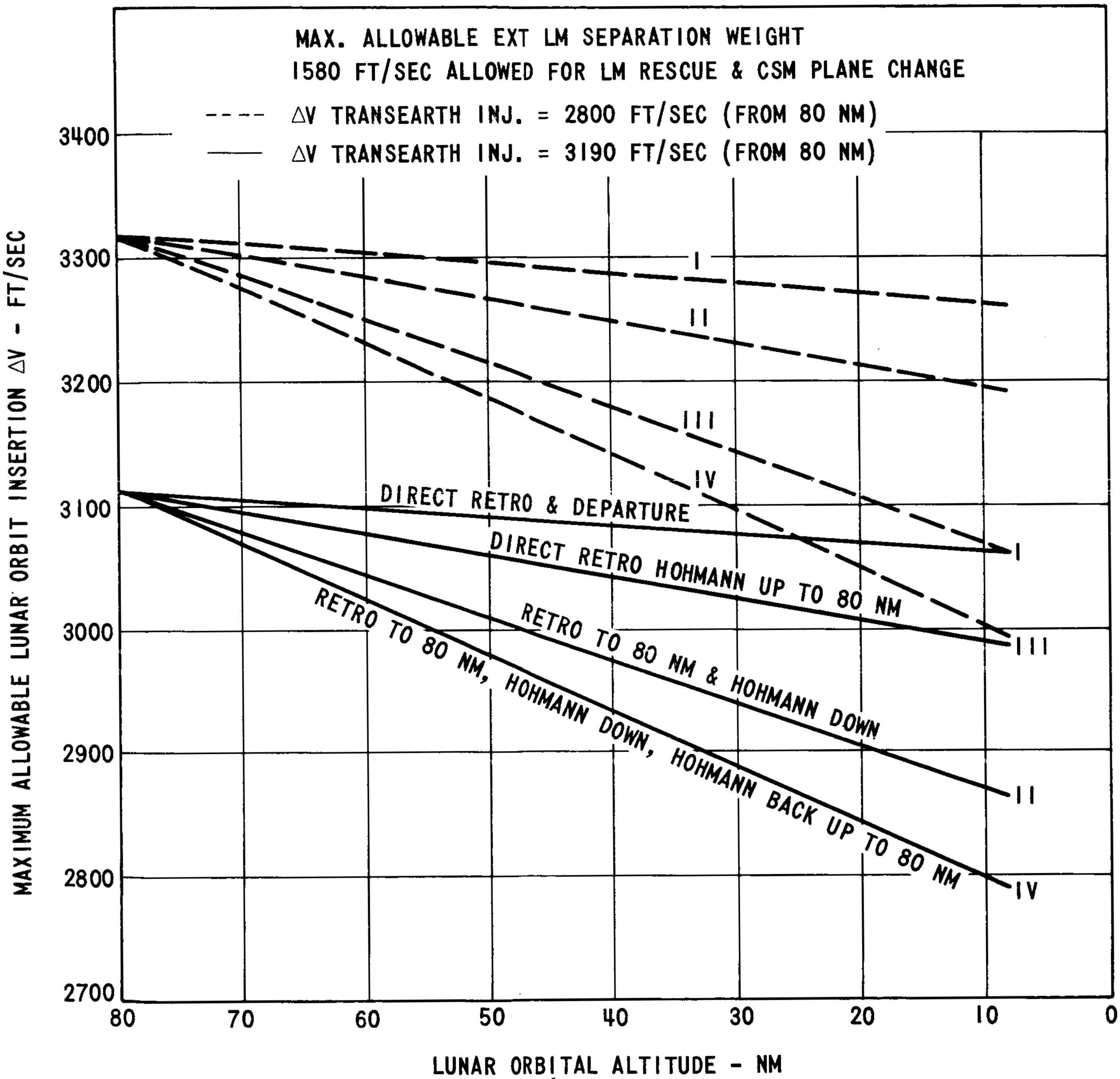

FIGURE 8 - MAXIMUM ALLOWABLE LUNAR ORBIT INSERTION ΔV FOR FULL BLOCK II SM PROPELLANT TANKS

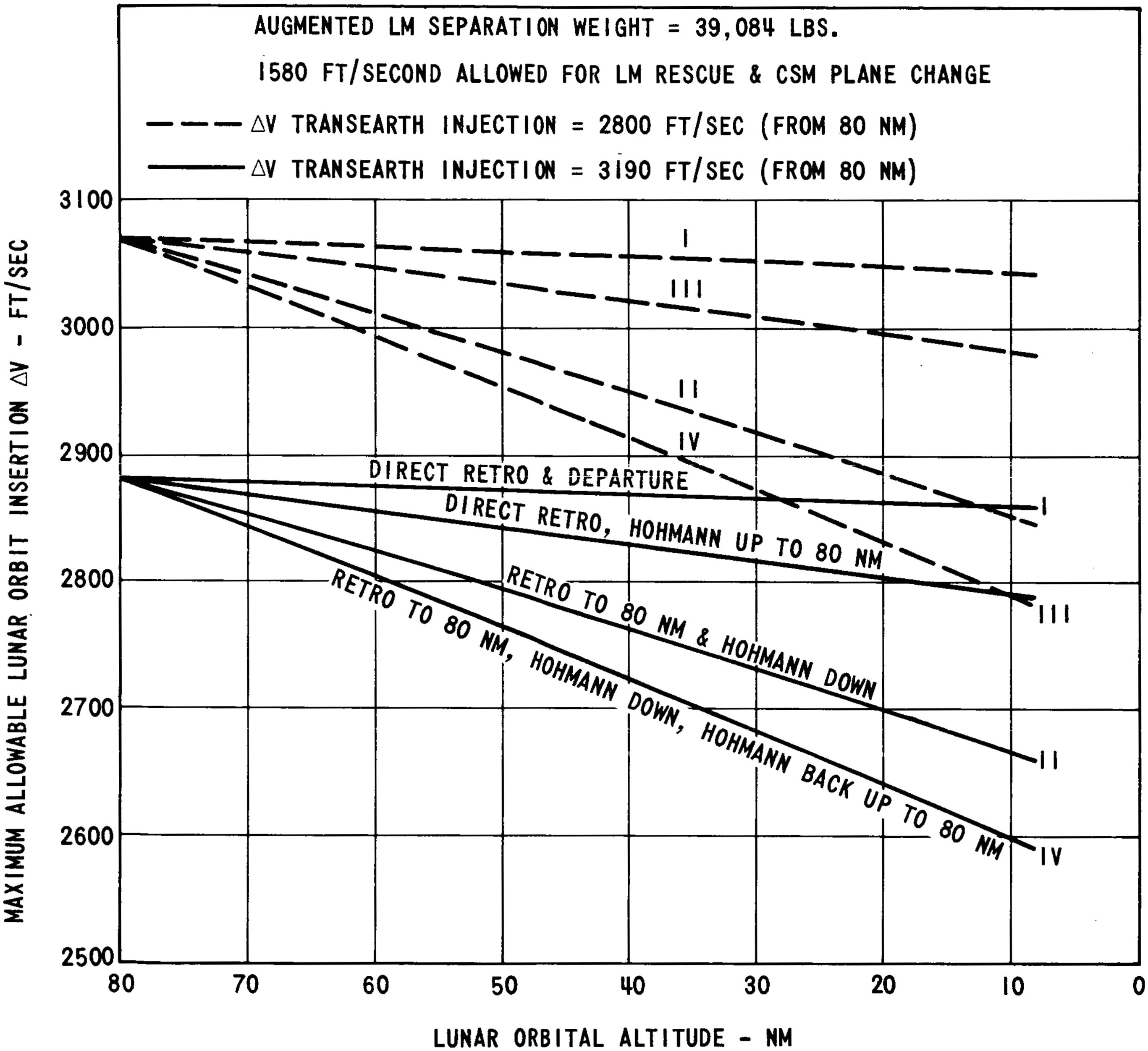

FIGURE 9 - MAXIMUM ALLOWABLE LUNAR ORBIT INSERTION ΔV FOR FULL BLOCK II SM PROPELLANT TANKS

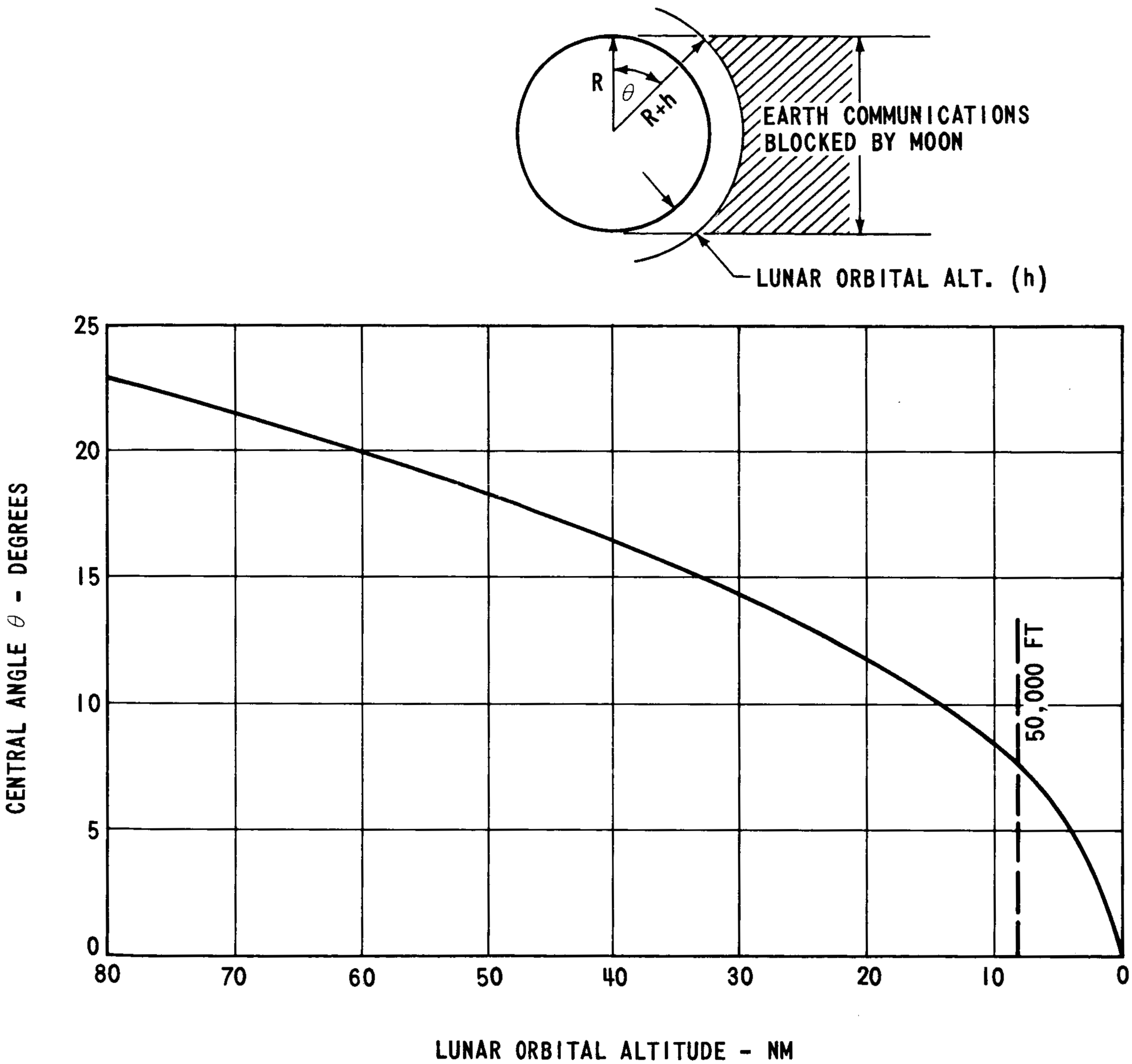

FIGURE 10 - VARIATION WITH ORBITAL ALTITUDE OF CENTRAL ANGLE (θ) TRAVELED BEYOND LIMB BEFORE EARTH COMMUNICATIONS ARE BLOCKED

APPENDIX A

SM ΔV PENALTY

The ΔV penalties associated with direct retro (LOI) and departure (TEI) from lunar orbital altitudes lower than the nominal 80 nm of Apollo are derived.

Assume a constant energy for the translunar flight trajectory:

(1) $$E = \frac{V^2}{2} - \frac{\mu}{R + h} \qquad \text{(expression for energy)}$$

As $R + h$ approaches ∞; $\frac{\mu}{R + h} \to 0$ and;

$$E = \frac{V_\infty{}^2}{2}$$, where V_∞ = hyperbolic excess velocity of the translunar trajectory

Substituting (2) into (1):

$$\frac{V_\infty{}^2}{2} = \frac{V^2}{2} - \frac{\mu}{R + h}$$

$$V^2 = V_\infty{}^2 + \frac{2\mu}{R + h}$$

$$V = \sqrt{V_\infty{}^2 + \frac{2\mu}{R + h}}$$

μ = lunar gravitational constant = 1.7313971×10^{14} ft^3/sec^2

R = lunar radius = 5.702395×10^6 ft

h = spacecraft altitude in feet

V = spacecraft velocity approaching the moon

Since circular orbital velocity = $\sqrt{\frac{\mu}{R + h}} = V_c$

(4) $$V = \sqrt{V_\infty{}^2 + 2V_c{}^2}$$

The ΔV required to retro from translunar flight to a circular orbit about the moon is the difference between the approach velocity (V) and the circular orbit velocity:

$$\Delta V = V - V_c$$

Or, substituting equation (4) for V:

$$\Delta V = \sqrt{V_\infty^2 + 2V_c^2} - V_c \qquad (5)$$

Using the nominal value for ΔV at an 80 nm altitude, equation (5) can be solved for V_∞ of the translunar flight trajectory. 2,957 ft/sec is the nominal ΔV, but allowances for midcourse correction and 3σ errors must be subtracted to obtain the theoretical ΔV requirement for retro.

$$\Delta V_{LOI} = 2{,}957 - 62(MCC) - 94(3\sigma) = 2{,}795 \text{ ft/sec}$$

$$2{,}795 = \sqrt{V_\infty^2 + 2(5{,}289)^2} - 5{,}289$$

$$5{,}289 = V_c \text{ at } 80 \text{ nm}$$

$$V_\infty = 3{,}067 \text{ ft/sec}$$

Using this value of V_∞ for translunar flight and substituting circular orbital velocities for different altitudes into equation (5) provides the retro requirement for each altitude of interest.

Example: for 50,000 ft altitude; $V_c = 5{,}486$ ft/sec

$$\Delta V_{LOI} = \sqrt{(3067)^2 + 2(5486)^2} - 5486$$

$$\Delta V_{LOI} = 2857 \text{ ft/sec}$$

Adding the allowances for MCC and 3σ errors:

$$\Delta V = 2857 + 62 + 94 = 3019 \text{ ft/sec}$$

∴ The ΔV penalty for retro to a 50,000 ft orbit instead of 80 nm = 3019 - 2957 = 62 ft/sec.

Results for altitudes between 50,000 ft and 80 nm are plotted in Figure 5.

The same procedure can also be applied for the transearth injection phase. The value of V_∞ for the trans-earth trajectory is computed as before using the nominal ΔV T.E.I. (less 3σ error allowance) in equation (5). With the value of V_∞ determined, the same procedures as above can be applied to determine the penalty associated with departure from lower orbits. Figure 5 also contains a plot of these penalties.

The nominal ΔV budget assumed for mission analysis purposes is as follows:

ΔV for lunar orbit insertion = 2957 ft/sec

ΔV for LM rescue and CSM plane change = 1580 ft/sec

ΔV for transearth injection = 3190 ft/sec

Since this is a nominal ΔV budget for an 80 nm lunar orbital altitude, it must be adjusted to include penalties of retro and departure from lower orbits as well as the requirements of the Hohmann transfers* of mission modes II, III and IV.

On this basis, the following ΔV budgets were developed for the four different mission modes included in the analysis:

MISSION MODE I

(RETRO INTO AND DEPART FROM LOW ORBIT)

LM Separation Altitude	ΔV TEI (ft/sec)	ΔV LM Rescue and CSM Plane Change (ft/sec)	ΔV LOI (ft/sec)
80 nm (nominal)	3,190	1,580	2,957
70 "	3,197	"	2,965
60 "	3,204	"	2,974
50 "	3,211	"	2,982
40 "	3,218	"	2,990
30 "	3,225	"	2,999
20 "	3,233	"	3,008
10 "	3,240	"	3,017
50,000 ft	3,242	"	3,019

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*Hohmann transfer ΔV requirements can be accurately estimated as the difference in circular orbital velocities at the two altitudes of interest.

MISSION MODE II

(RETRO TO 80 NM AND HOHMANN DOWN TO LOW ORBIT)

LM Separation Altitude	ΔV TEI (ft/sec)	ΔV LM Rescue and CSM Plane Change (ft/sec)	ΔV LOI + Hohmann Down (ft/sec)
80 nm (nominal)	3,190	1,580	2,957
70 "	3,197	"	2,983
60 "	3,204	"	3,010
50 "	3,211	"	3,037
40 "	3,218	"	3,064
30 "	3,225	"	3,092
20 "	3,233	"	3,120
10 "	3,240	"	3,149
50,000 ft	3,242	"	3,154

MISSION MODE III

(RETRO DIRECTLY TO LOW ORBIT,
HOHMANN UP TO 80 NM, AFTER LM SEPARATION)

LM Separation Altitude	ΔV TEI (ft/sec)	ΔV LM Rescue + CSM Plane Change + Hohmann up to 80 NM (ft/sec)	ΔV LOI (ft/sec)
80 nm (nominal)	3,190	1,580	2,957
70 "	"	1,606	2,965
60 "	"	1,633	2,974
50 "	"	1,660	2,982
40 "	"	1,687	2,990
30 "	"	1,715	2,999
20 "	"	1,743	3,008
10 "	"	1,772	3,017
50,000 ft	"	1,777	3,019

MISSION MODE IV

(RETRO TO 80 NM, HOHMANN DOWN TO LOW ORBIT,
HOHMANN BACK UP TO 80 NM AFTER LM SEPARATION)

LM Separation Altitude	ΔV TEI (ft/sec)	ΔV LM Rescue + CSM Plane Change + Hohmann up to 80 nm (ft/sec)	ΔV LOI + Hohmann Down from 80 nm (ft/sec)
80 nm (nominal)	3,190	1,580	2,957
70 "	"	1,606	2,983
60 "	"	1,633	3,010
50 "	"	1,660	3,037
40 "	"	1,687	3,064
30 "	"	1,715	3,092
20 "	"	1,743	3,120
10 "	"	1,772	3,149
50,000 ft	"	1,777	3,154

APPENDIX B

Determination of SM propellant requirement for a lunar landing mission.

Let:

LM = LM separation weight (including crew)

CSM = weight of CSM (including crew) less weight of usable propellant

SMP = Service Module usable propellant weight

EX_1 = weight of CSM expendables lost prior to lunar orbit insertion

EX_2 = weight of CSM expendables lost in lunar orbit

Crew = weight of LM crew transferred from CSM

R_1 = mass ratio associated with ΔV for T.E.I.

R_2 = mass ratio for ΔV requirement of maneuvers accomplished by CSM without the LM or its crew attached. (LM rescue, CSM plane change, and Hohmann)

R_3 = mass ratio associated with ΔV for lunar orbit insertion + maneuvers (Hohmann transfer) with LM and crew attached

$$R = \exp \frac{\Delta V}{g_o Isp}$$

MISSION CALCULATION

Spacecraft weight just after transearth injection

$= CSM - (EX_1 + EX_2)$

Spacecraft weight just prior to transearth injection

$= [CSM - (EX_1 + EX_2)] [R_1]$

SM propellant used for transearth injection

(1) $= [CSM - (EX_1 + EX_2)] [R_1 - 1]$

Spacecraft weight just after LM rescue,

CSM plane change and Hohmann up

$= [CSM - (EX_1 + EX_2)] [R_1] - Crew*$

Spacecraft weight just prior to these maneuvers

$= \Big[[CSM - (EX_1 + EX_2)] [R_1] - Crew \Big] [R_2]$

SM propellant used for these maneuvers

(2) $= \Big[[CSM - (EX_1 + EX_2)] [R_1] - Crew \Big] [R_2 - 1]$

Spacecraft weight just after lunar orbit insertion

$= \Big[[CSM - (EX_1 + EX_2)] [R_1] + Crew \Big] [R_2] + EX_2{}^{**} + LM$

Spacecraft weight just prior to lunar orbit insertion

$= \Big[\Big[[CSM - (EX_1 + EX_2)] [R_1] - Crew \Big] [R_2] + EX_2 + LM \Big] [R_3]$

SM propellant used for lunar orbit insertion

(3) $= \Big[\Big[[CSM - (EX_1 + EX_2)] [R_1] - Crew \Big] [R_2] + EX_2 + LM \Big] [R_3 - 1]$

Total SM propellant required for the mission is equal to the sum of equations (1), (2), and (3).

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*LM crew not on board the CSM during LM rescue maneuvers

**Expendables lost in lunar orbit (EX_2) are on board at the time of lunar orbit insertion.

Adding and simplifying:

Total SM propellant required for the mission

(4) $$= [CSM - (EX_1 + EX_2)] [R_1R_2R_3 - 1] - Crew [R_2R_3 - 1] + [LM + EX_2] [R_3 - 1]$$

Equation (4) was used to determine the total SM propellant requirement for the different mission modes analyzed. The following numerical values were used:

CSM = 23,562 lbs (baseline value)

LM = Maximum LM separation weight from Figure 3 for each altitude

Crew = 589 lbs (baseline)

EX_1 = 261 lbs (from DRM IIA)

EX_2 = 200 lbs (from DRM IIA)

SM Isp = 313 sec

ΔV Budget = as tabulated in Appendix A for each mission mode

APPENDIX C

DETERMINATION OF MAXIMUM ALLOWABLE ΔV BUDGET FOR BLOCK II SM PROPELLANT CAPACITY (38,986 LBS)

The SM propulsion requirements can be reduced to three propulsion events: (1) lunar orbit insertion, which is performed with the full spacecraft weight; (2) maneuvers in lunar orbit, which are performed without the weight of the LM or its crew; and (3) transearth injection, which is performed with the CSM and the LM crew weights. If the expression for SM propellant developed in Appendix B is set equal to the present Block II SM tank capacity (38,986 lbs) and two of the three SM propulsion requirements (ΔV's) are held fixed,* it is possible to solve for the remaining ΔV requirement. Figure 8 is a plot of the results of this type calculation. The transearth injection ΔV requirements used were the nominal 3,190 ft/sec for one set of results and 2,800 ft/sec to illustrate the effect of this type change. The nominal ΔV requirements for LM rescue and CSM plane change were held at 1,580 ft/sec in all cases. Figure 8 shows the maximum allowable ΔV requirement for lunar orbit insertion for each of the four mission modes for the two values of transearth injection ΔV.

The following tabulation of results from Figures 8 and 9 should provide some feel for the range of ΔV requirements allowable with the existing SM tank capacity, the maximum allowable Extended LM separation weight (Figure 2), and a LM separation altitude of 50,000 ft. For each of the Extended LM missions tabulated below, the total spacecraft injection weight is 99,083 lbs (32,685 lbs Ext. LM (less crew); 23,562 lbs CSM; 38,986 lbs SM propellant and 3,850 lbs SLA). The Augmented LM missions shown on Figure 9 require a Saturn V capability of 104,893 lbs.

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*The fixed ΔV requirements assumed and the results obtained must be modified to allow for maneuvers associated with the individual mission modes; i.e., direct retro and departure from lower orbits or Hohmann transfers.

Mission Mode	ΔV TEI (ft/sec)	ΔV LM Rescue and CSM PC (ft/sec)	Max. Allowable ΔV for LOI (ft/sec)	
			Ext. LM	Aug. LM
I	3,190	1,580	3,062	2,858
	2,800	"	3,262	3,042
II	3,190	"	2,865	2,661
	2,800	"	3,064	2,845
III	3,190	"	2,986	2,788
	2,800	"	3,193	2,979
IV	3,190	"	2,789	2,591
	2,800	"	2,996	2,782

The type information from Figure 8 should be useful for mission planning purposes. The allowable ΔV budget will enable an assessment of the required relaxation to the nominal Apollo requirements. Such things as the necessity of non-free return flights, launch opportunities, launch windows and trip times could be readily determined for missions to specific landing sites.

Figure 9 represents the same type information except that the LM separation weight has been increased to the Augmented LM value (38,495 lbs LM + 589 lbs Crew = 39,084 lbs). The increased LM separation weight reduces the maximum allowable ΔV for the retro maneuver by about 200 ft/sec for mission conditions corresponding to the Extended LM.